Cassini Spacecraft Design

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ABSTRACT

The Cassini spacecraft will explore the planet Saturn and its rings and moons with an orbiter and atmospheric entry probe, both of which have a sophisticated set of science instruments. The spacecraft design is responsive to mission and science objectives, and is influenced by technical and programmatic constraints e.g., a cost cap, fixed schedule, space environment, and interfaces to other fixed systems like the launch vehicle and ground system The spacecraft design must also consider the limited post-launch resources allocated to do the flight operations. This paper presents an overview of the spacecraft system design with emphasis given to the orbiter and only a high level summary of the probe.

Keywords: Cassini, spacecraft, Huygens, Saturn, Titan

1. Introduction

1.1 The Mission

The mission consists of a 7 year cruise followed by 4 year orbital operations. The spacecraft will be launched from Cape Canaveral in October 1997. The Titan IV/Centaur launch vehicle places the S/C on a trajectory that has several fly-bys of inner planets before encountering Saturn, Specifically, the spacecraft will fly-by, and obtain gravity assists from Venus (April 1998), Venus again (June 1999), Earth (August 1999), and Jupiter (December 2000). The spacecraft encounters Saturn in July 2004, and executes a retrograde maneuver, SOI, which allows the gravity of Saturn to capture the spacecraft into orbit. 130 days after SOI the orbiter releases the Huygens Probe on a trajectory to intersect Saturn's moon Titan, where 21 days later it will descend through the atmosphere and relay science data to the. Orbiter. The radio link between the Probe and the Orbiter will provide for an approximate 3 hour Probe mission. The Orbiter capture the probe data and relays it to earth, and then begins its own 4 year exploration of the Saturnian system consisting of the planet, its rings, and its moons.

1.2 The Spacecraft System

The Spacecraft is composed of the Orbiter, the Huygens Probe, and the launch vehicle adapter. The Orbiter consists of 12 scientific instruments plus Orbiter engineering and Probe receiving equipment. The Orbiter instruments are listed below and consist of four Optical Remote Sensing Instruments, two Microwave Remote Sensing Instruments, and six Fields / Particles / Waves instruments. The Huygens probe consists of six science instruments plus engineering equipment, the scientific instruments are listed below.

orbiter **Instruments**

Optical Remote Sensing
- Imaging Science Subsystem (1SS)
- Visible and Infrared Mapping Spectrometer (VIMS)
- Ultraviolet Imaging Spectrograph (UVIS)
- Composite Infrared Spectrometer Subsystem (CIRS)

Microwave Remote Sensing
- Radar Subsystem (RADAR)
- Radio Frequency Instrument Subsystem (RFIS)

Fields / Particles / Waves
- Cassini Plasma Spectrometer Subsystem (CAPS)
- Cosmic Dust Analyzer Subsystem (CDA)
- Ion and Neutral Mass Spectrometer Sub.(INMS)
- Magnetometer Subsystem (MAG)
- Magnetospheric Imaging Instrument Sub.(MIM 1)
- Radio and Plasma Wave Science Sub.(RPWS)

Huygens Probe **Instruments**

- Aerosol Collector Pyrolyzer
- Descent Imager / Spectral Radiometer
- Doppler Wind Experiment
- Gas Chromatograph / Mass Spectrometer
- Huygens Atmospheric Structure Instrument
- Surface Science Package

2. Orbiter System Design Requirements and Drivers

The key drivels on the Orbiter design arc:

a. Accommodation of a large and sophisticated science payload,
b. The trajectory inducting the range of spacecraft . earth and spacecraft - sun distances.
c. Accommodation of the Huygens Probe,
d. The Space environment.
c. Single fault tolerance, derived from the long lifetime.
f. The need to build in operability such that the spacecraft can be operated by a small flight team

The Orbiter design response to these key drivers is summarized in Table 1.

Key Design Drivers	Design Response
1. Large, sophisticated Science Payload	
fields of view	a fields and particles pallet and a remote sensing pallet
pointing	stellar reference units and reaction wheels for high performance pointing knowledge (1 mrad) and control (2 mrad)
contamination - optical	the configuration establishes spatial separation between the main engine and the science optics; plume control; low gassing materials
contamination - magnetic	strict electromagnetic emission requirements and testing; magnetic shielding; configuration of MAG sensors on a boom
contamination - RFI	strict radio frequency emission requirements and testing; Faraday cage containment; RPWS antennas deployable
science data generation rate from 4 kbps to 410 kbps	solid state recorder; independent operation of record & playback; selectable data rates
2. Trajectory	
Orbiter-sun distances range from 0.67 AU (Astronomical Units) to 10.2 Au	Thermally designed for the "no-sun" case, use of the HGA as a "" sun.shield within 2,7 AU
Orbiter-earth distances range from <.1 AU (cKuisc)to>11 AU	a 4m high gain antenna, X-band telecommunication system; selectable data rates; autonomous fault protection

3. Huygens Probe	
power	?.30 W for checkout, 70 W for probe relay
data return	redundant storage for Probe relay data; Orbiter memory stores Probe command sequencing
structural, thermal, Probe release mechanism	accomm I iodated
Probe release	Orbiter aligns the Probe and releases imparting a spin of 5 rpm and a separation velocity Of 0.3 m/s; regains attitude control of Orbiter 120 minutes after Probe release.
Probe radio relay	capability to point the HGA al a surface feature of Titan
4. Space Environment	
radiation	use of radiation hardened parts; shielding
"single event upset"	error detection and correction circuitry on computer memory; onboard copies of software maintained fOr autonomous backup.
micrometeoroid	shielded by multi-layer blanketing and structure;the main engine has a retractable cover
5. Fault Tolerance	
Single fault tolerance - hardware	block redundant architecture; balanced power bus; fault isolation at interfaces
Single fault tolerance - autonomy	onboard software to perform health checks and capability to establish a minimum safe state Orbiter
Single fault tolerance - key mission events	onboard software 10 perform health checks and to perform configuration changes, if necessary, to ensure completion of key mission success required events.
6. Small Flight Team	defined set of Operational Modes; autonomous fault tolerance; sequence storage of 150 kwords; power margin

'fable 1 Key Orbiter Design Drivers / 1 Design Responses

3.0 Orbiter System Design

3.1 System Architecture

The Orbiter is comprised of a Power System, Guidance and Control System, Information System, Telecommunication System, Science Payload, Thermal Control System, and a distributed Fault Protection System. It has an information interface with the Ground operations team, a structural and information interface with the launch vehicle, and sensor interfaces with the space environment. Figure 1 is a functional block diagram of the Orbiter.

A key architectural design feature. is the implementat iou of Operational Modes. The Orbiter is not capable of providing sufficient resources to meet all combinations of science payload operation. Spacecraft pointing, data rates, and power must be allocated, Operational Modes define Orbiter states which have a primary science goal with an associated science instrument suite and the necessary engineering support. The implementation of a fixed number of Operational Modes (using a pre-defined transition architecture) is also a major contributor to the Orbiter compatibility with a small operations team by reducing sequence design and analysis efforts.

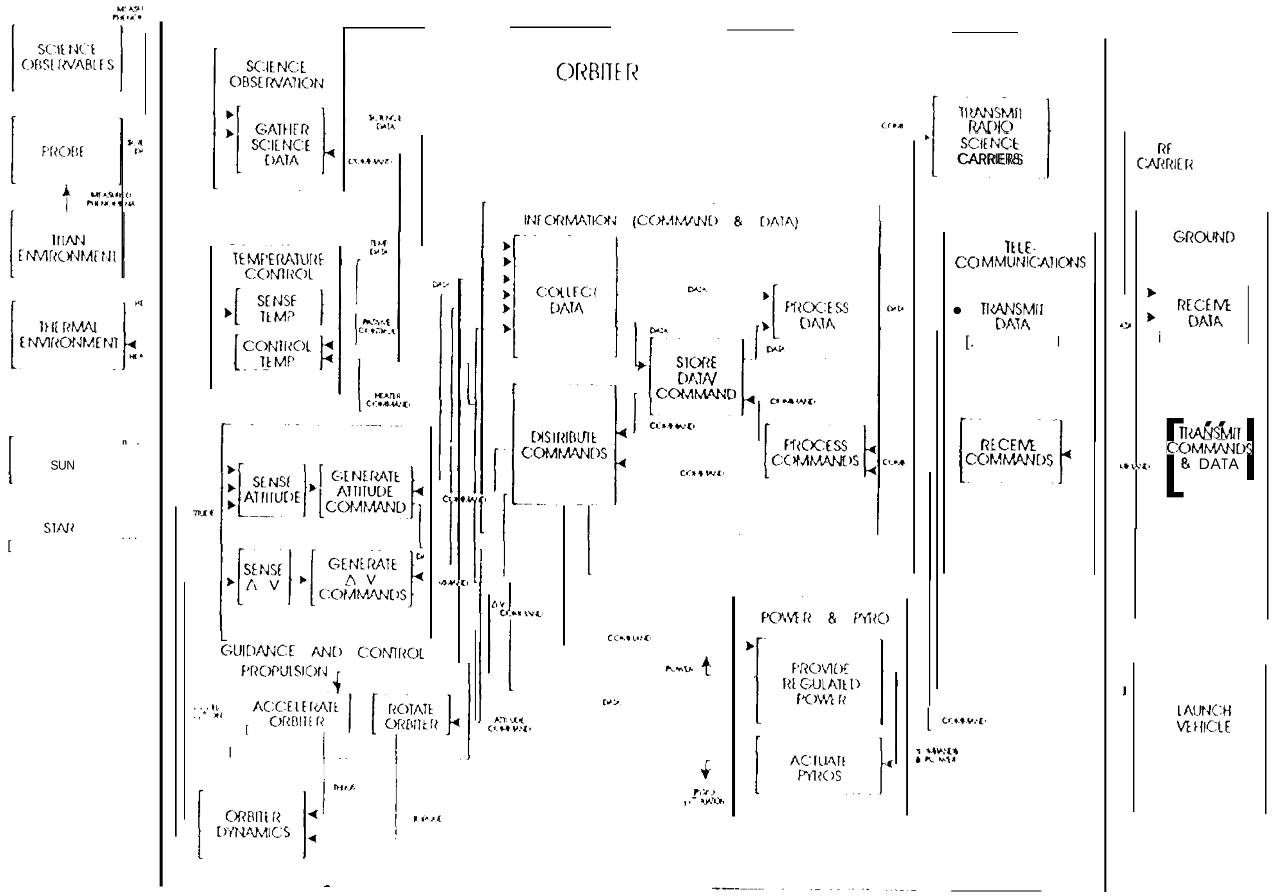


Figure 1. Orbiter Functional Block Diagram

3.1.1 System Configuration

Figure 2 shows the Orbiter in the Cruise configuration. The configuration is comprised of a stack consisting of the High-Gain Antenna (HGA), the Upper Equipment Module (UEM), the ~'repulsion Module Subsystem (PMS), and the Lower Equipment Module (LEM).

The UEM consists of a 12 bay electronics bus (configured to form a Faraday cage), the science Fields and Particles Pallet (FPP), the science Remote Sensing Pallet (RSP), the upper shell structure, the MAG boom, and the shunt radiator, The PMS is a structural assembly comprised of the propulsion subsystem. The LEM supports the PMS on its upper end and attaches to the LSA (not shown in figure 1) on its lower end. It supports three RTGs, the three fixed Reaction Wheel Assemblies (RWAs), LGA-2, separation hardware, and the MEA cover assembly (not shown in figure 1). The LSA mounts 10 the launch vehicle adapter.

In the Launch configuration the Orbiter is attached to the Launch vehicle and all deployables are stowed or latched. In the Cruise configuration the Orbiter is separated from the Launch vehicle, the RPWS antenna and the Magnetometer boom are deployed, and the articulated Reaction Wheel Assembly is un-latched. This Orbiter has a few deployable elements compared to previous interplanetary spacecraft; they were held to a minimum to reduce cost and to increase reliability.

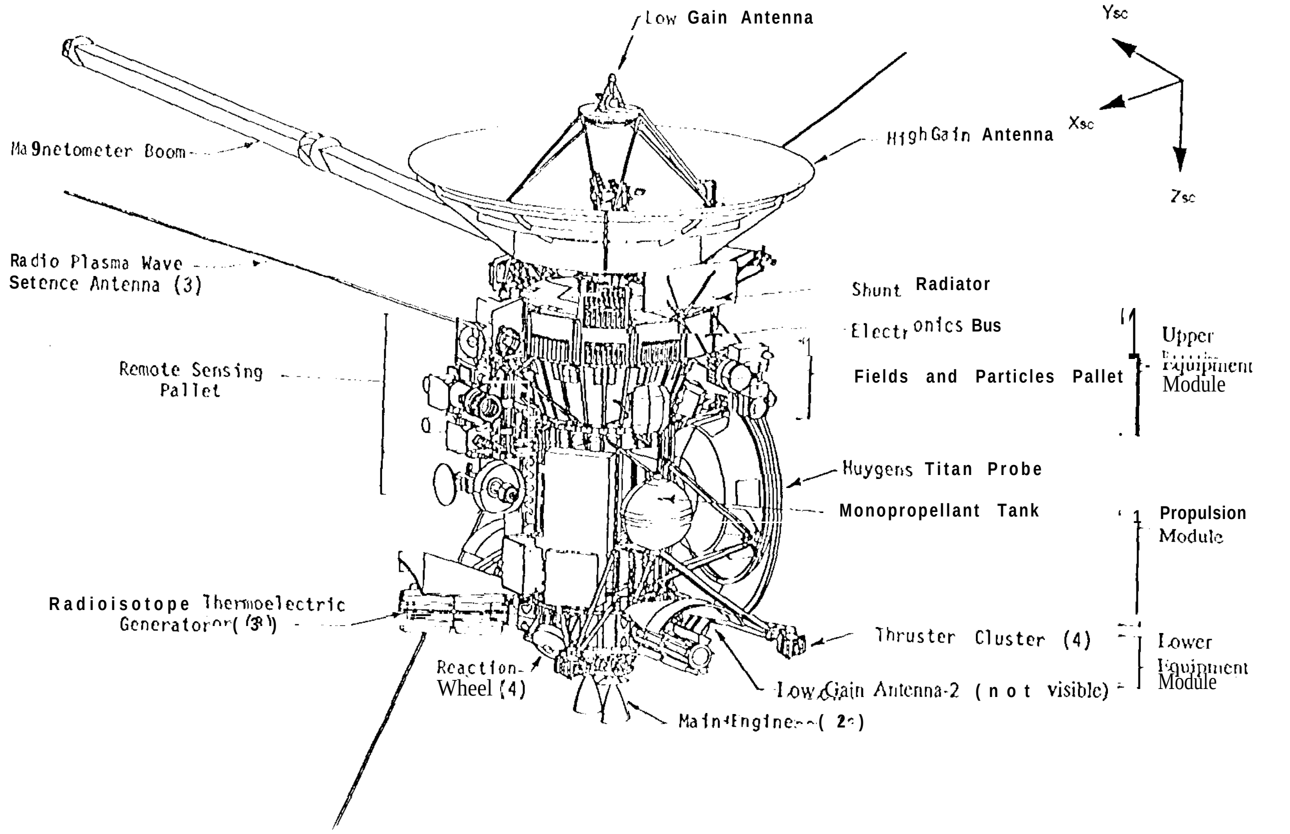


Figure 2. Cassini Spacecraft

Height	6.8 m (22.3 ft)
Diameter (excluding MAG boom)	4 m (13.1 ft)
Mass	
Orbiter	2150 kg
Huygen Probe	343 kg
probe support equip	30 kg
launch adapter	165 kg
propellants	3132 kg
total	5820 kg (12,346 lb.)

Table 2 Spacecraft Physical Characteristics

This configuration meets the payload requirements. It provides the science instruments with unobstructed fields of view for both the sensing optics and heat radiation. The RTGs are located away from the bays containing the science sensors and the science / engineering electronics assemblies to minimize the effects of the ionizing and thermal RTG radiation, The RWAS are located on the LEM to minimize mechanical and magnetic disturbances to the science sensors. The electronics are located in the electromagnetically shielded bay assembly to contain the emission energy that would otherwise interfere with the RPWS operation The low gain antennas are locate.d to provide omnidirectional telecommunication coverage. The main engine assembly is located below the LEM so that the engine plume does not have a direct path to the science sensors located on the instrument pallets.

This configuration allows the HGA to provide several functions: sufficient gain to allow high rate communication with Earth at X-band; a high gain/ narrow beamwidth aperture for the Ku-band RADAR, a transmit antenna for S-band and a transmit/receive antenna for Ka-band radio science, a high gain S-band receive antenna for receipt of the Huygens Probe data; a structural mount for a low gain antenna (LGA-1); and a sun shade for the majority of the Orbiter.

3.1.2 Power System

The Orbiter uses RTGs as the power source. The radioactive decay of the plutonium pellets within each RTG produces heat which is converted into electrical energy by an a nay of thermocouples made of silicon-germanium junctions. The electrical power available is slightly over 800 watts initially and decays at approximately one to two percent per year with approximately 640 watts available at the end of mission.

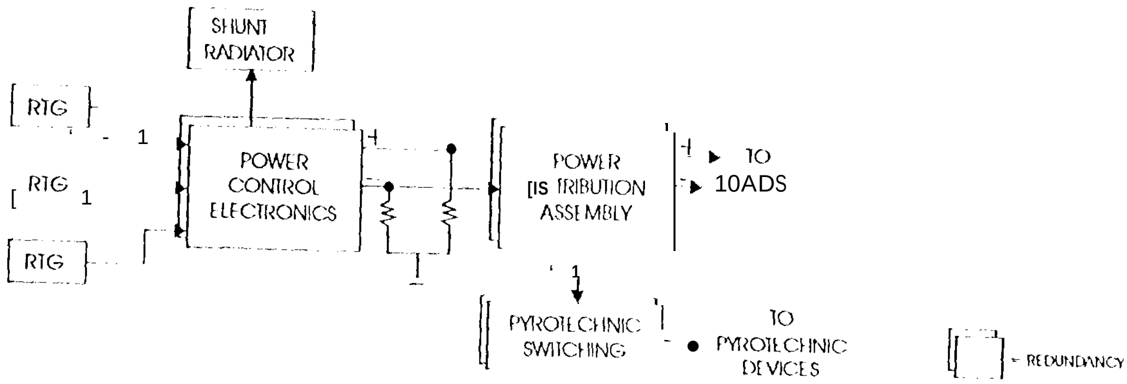


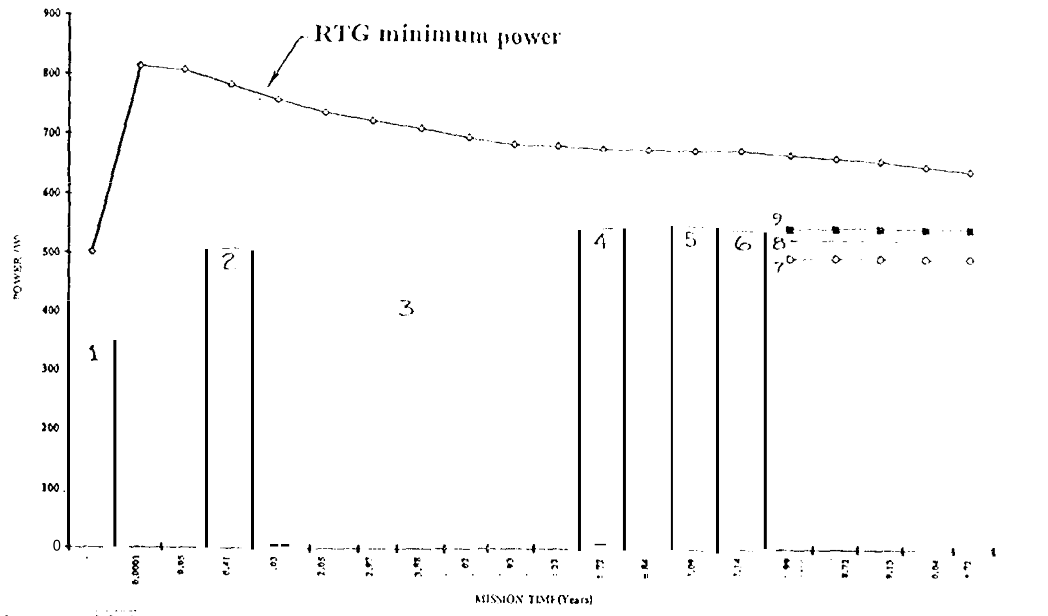
Figure 3 Power System Functional Block Diagram

Figure 3 is a functional block diagram of the power system. The electrical power output of the RTGs is regulated and conditioned by the Power Control Electronics (PCE) and supplied as 30 Vdc. Excess electrical power is dissipated by the shunt radiator as heal. The loads arc controlled by the Power Distribution Assemblies (PDAs), which, combined, house 192 Solid-State Power Switches (SSPSs), which switch the electrical loads to the spacecraft power bus. The SSPS design provides ramped turn-on, quick acting, over-current clamping and trip protection, and switch state and current telemetry.

The electrical configuration of the power system provides a balanced system, meaning that the chassis / structure of the spacecraft is isolated through resistors from both rails of the power bus. This configuration eliminates the possibility that a single short can disable the power system. All spacecraft loads, consequently, arc designed to operate properly when either side of the power bus is grounded (by high or low resistance) to chassis.

The Power System dots not have the capability to support the entire science payload operating in peak power mode. This has resulted in the definition of Operating Modes as described in section 3.1 Table 3 illustrates that the design maintains sufficient operating margin through end of mission for a set of the worst case modes.

The Pyrotechnic Switching Unit (PSU) provides a energy storing capacitor bank and power switching for fining electro-explosive (pyre) devices. The capacitor bank is capable of supplying up to 25 Amps for pyro initiation. The PSU has the capability to initiate 34 separate pyro events, each event firing onc to three pyrotechnic devices. Pyrotechnic devices are used to separate the Orbiter from the Launch Vehicle, actuate propulsion isolation valves, release the Magnetometer boom and Langmuir probe, jettison science instrument covers, unlatch the articulated RWA, and separate the Huygens Probe from the Orbiter.



1- Launch Mode
2- Decontam.
3 - Cruise Mode
4- SOI Ignition Peak
5 . Final Probe Checkout
6 - Probe Entry
7 Remote Sensing, INMS, Fields and Particles
8- Remote Sensing, RW unload :CBH Warmup
9- Remote Sensing, RADAR, INMS & CAPS

“J’able 3. Peak Powei vs Opcraling Modcs and Mission Timc

3.1.3 Guidancc and Control Systcm

The Guidance and Control systcm is required to maintain spacecraft attitude, perform changes to the trajectory, execute a series of basebody pointing slews and holds to create an image mosaic, and turn the spacecraft to point either the HGA or the Remote Sensing instrument axis toward targets Note. that the tar.gels selected may be a body moving in inertial space (e.g., a RADAR scan of the surface of Titan may require the Guidance and Control system to maintain the HGA axis pointed at a fixed position of Titan’s surface while both Titan and the Orbiter move through inertial space). Target vectors arc propagated onboard by the Guidance and Control computer based on g[omd-supplied equations.

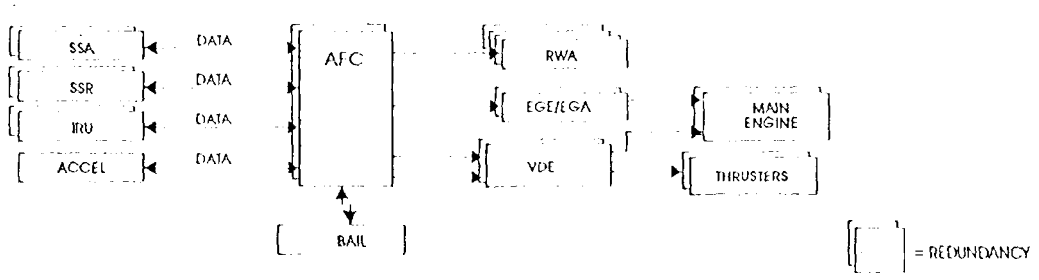


Figure 4 Guidance and Control Functional Block Diagram

Figure 4 is a functional block diagram of the Guidance and Control System This architecture uscs software algorithms resident on the flight computer (AFC) to process the data available from the sensors (giving position / turn rate/ acceleration information) to determine the necessary commands to actuate torque or acceleration control of the spacecraft. The sensors used arc the Sun Sensor Assembly (SSA), the Stellar Reference Unit (SRU), and an accelerometer (ACC).

Torque is applied by either the use of the Reaction Wheel Assemblies (RWAs) Or thrusters. Acceleration is applied by the use of either the thrusters or the Main Engine Assembly (MEA).

The Sun Sensor Assembly (SSA) and the Stellar Reference Unit (SRU) both provide Orbiter position The SSA provides for sun acquisition and coarse (1 ,5°) 2 axis attitude information At least one SSA will be powered on at all times during the mission. The SRUs are 15° square field-of-view star imagers used (o locale reference stars relative to the spacecraft axis. The SRU provides the AFC with up to 50,000 pixels of information per second. Software algorithms resident on the AFC establish and maintain stellar reference by comparing incoming pixel frames to an onboard catalog of approximately 5000 stars. The SRU allows the determination of attitude position to an accuracy of better than 1 milli-radian for attitude rates below 0.3° / sec.

The Inertial Reference Unit (IRU) provides Orbiter rate information to the AFC which allows the maintenance of attitude knowledge while the spacecraft is performing a turn and during main engine propulsive maneuvers. Each IRU has a set of four solid-slate Hemispherical Resonator Gyroscopes (HRGs). The inertial sensitive element in each HRG is a fused silica shell, the hemispherical resonator. The HRG is excited to produce standing waves in the shell and when the shell is rotated about its axis, the oscillating mass elements experience forces that cause the standing wave to precess (i.e., rotate) with respect to the shell. The precession angle is a constant fraction of the angle through which the shell has rotated, and this allows precise measurement of angular motion in the axis of the HRG. The IRU meets all performance requirements at spacecraft rates of up to 20/second and remains usable, although with lower accuracy, at rates up to 15°/sccord.

The ACC provides the AFC with the acceleration along the Z axis, A main engine burn will result in an acceleration of about 20 - 30 milli-g. The ACC provides a resolution of (J. ? milli-g. The software in the AFC terminates main engine burns when the accelerometer indicates that the requested velocity has been achieved and the minimum burn duration (as calculated by the operations team) has been met or when the maximum burn duration was been achieved.

There are two propulsion systems on the Orbiter: a bipropellant (nitrogen tetroxide ((N_2O_4)) / monomethyl hydrazine (($N_2H_3CH_3$))) main engine system and a monopropellant (hydrazine ((N_2H_4))) thruster system The bipropellant rocket engine system, implemented with redundant 445 Newton thrust engines and gimbal actuators, is used for large trajectory changes. The largest of the maneuvers, Saturn Orbit Insertion (SOI), will require a delta-V of approximately 600 m/s equaling to a burn duration of about 90 minutes. The monopropellant system uses sixteen 1 Newton thrusters located around the Orbiter on four thruster clusters and is used for performing attitude control, small trajectory changes, and reaction wheel desaturation.

The RWAs provide primary attitude control during the science observation period at Saturn.. During the mission cruise phase, the pointing requirements will be adequately served by the thrusters. Once the spacecraft reaches Saturn however, the scientific observations require higher pointing accuracy and stability, as well as frequent spacecraft repositioning. This is accomplished by the RWAS which allow pointing control to an accuracy of 2 milli-radians. RWAS arc electrically powered wheels mounted in three orthogonal axes aboard the spacecraft. Driving the wheel speed up or down causes a torque to bc applied to the spacecraft. Excess momentum tends to build up in the system over time, due to the torque applied by the solar wind as well as internal RWA friction, and this is removed try the thrusters.

The VDE provides a command and telemetry interface between the AFC and the main engine, the sixteen hydrazine thrusters, and certain heaters and isolation valves.

The BAIL is a non-volatile memory storage device that is capable of loading the AFCs in the event of a fault that causes the loss of the software resident both in the AFC RAM and the backup copies stored on the SSR This allows restoration of an acceptable spacecraft attitude and thermal slate.

3.1.4 Information System

The Information System receives command loads sent from the operations team through the Telecommunication System, stores and executes command sequences, distributes commands to the spacecraft subsystems, collects and stores telemetry from the spacecraft subsystems, sends telemetry to the Telecommunication system for transmission to the operations team, and hosts the software algorithms which perform the Fault Protection services.

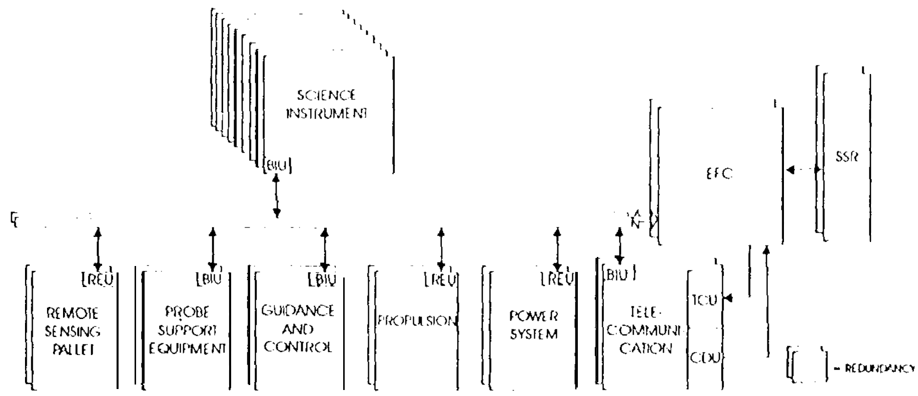


Figure 5 Information System Functional Block Diagram

Figure 5 is a functional block diagram of the Information System. This architecture uses the software resident on the flight computer (EFC) to collect/send data from/to the Orbiter and Probe support equipment. The EFC is based on the MIL-STD-1750A processor and is programmed in Ada . The Bus Interface Units (BIUs) and Remote Engineering Units (REUs) are remote terminals embedded in spacecraft subsystems, designed according to the MIL-STD-1553B data interface specification (modified to lower power requirements). The BIUs, located in the science instruments, the Telecommunication System, the Guidance and Control System, and the Probe Support equipment operate under the direction of the EFC extracting data from the host for transfer to the EFC and transferring data from the EFC into the host. REUs are remote terminals located in the Power System, the Propulsion module, and the Remote Sensing Pallet where they operate under the direction of the EFC to collect analog (e.g., voltage, temperature, pressure) and level discrect (e.g., switch status) data and to provide a command interface from the EFC to the host. The telemetry and command statistics are summarized in Table 4.

	number telemetry channels	number of command stems
Orbiter - science	1515	651
Orbiter - engineering	5525	462
Huygens Probe	1900	17
total	8940	1130

Table 4 Telemetry and Command Statistics

Each of the two Solid State Recorders (SSR) is capable of storing 2 billion bits (2 Gbits) of information. The SSR is used to store science data for those cases where the downlink is not available or the downlink rate is less than the science data generation rate. The SSR is also used to store backup copies of the software used in the Information System, the Guidance and Control System, and some of the science instruments. Due to the critical nature of the Probe data, the information System will record this data redundantly on both SSRs.

The Information System collects, stores or downlinks telemetry according to define.d telemetry modes. Mission requirements have led to the definition of 27 telemetry modes, summarized in Table 5. Programmatic considerations have led to the pre-launch development of 8 of the.se modes which will be used in integration and test activities, or in early cruise operations. The remaining modes are developed before launch as well, but will be uplinked to the spacecraft when they are needed.

Telemetry 'Mode Class	Record rate 10 SSR (kbps)	Playback rate from SSR (kbps)	Downlink rate
Realtime Engineering 4 modes	engineering: 1.6 science: O		20 bps -16.6 kbps
Engineering Playback & Realtime 1 mode	engineering: 1,6 science: O_.	'.034	4-0 bps
Science & Engineering Record 10 modes	engineering: 1,6 science: 4 - 4 1 5		
Realtime Engr + Science Playback 8 modes	engineering: 1.6 science: 4-415	".012. -.142	14.2 kbps - 166 kbps
Probe Checkout 1 mode	engineering: 1.6 Probe: 23.2		24.8
Probe Op 1 mode	TBD " "		TBD
SAF Checkout 2 modes	multiple	multiple	142, 249

Table 5 Telemetry mode.s

3.1.5 Telecommunication System

The Telecommunications System operates at X band (8.4 GHz downlink, 7.? GHz uplink) and communicates with the. Deep Space Network (DSN) 34 m and 70 m antennas. This system is designed to be capable of transmitting approximately 4 Gbits of data per 9 hour downli nk at Saturn Two low gai n antennas provide the capability for nearly omnidirectio nal coverage. This low gain coverage meets the requireme nt for commun ication capability during the cruise phase(while the HGA is sunpointed) and for fault-related attitude recovery responses.

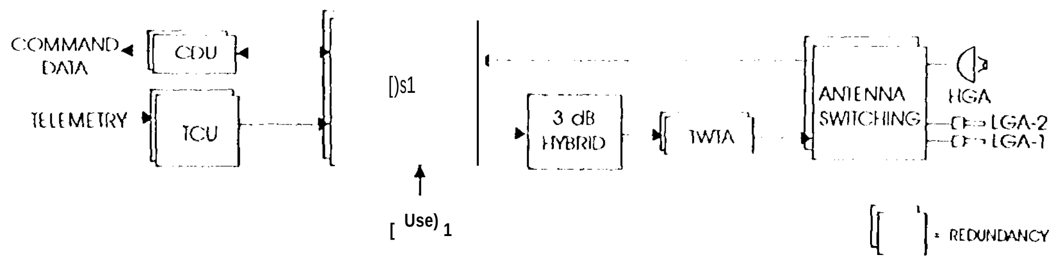


Figure 6 Telecommunicat ion Functional Block Diagram

Figure 6 is a functional block diagram of the telecommunication system. Telemetry data from the Information System enters the Telemetry Control Unit (TCU) where it is conditioned The TCU feeds this to the Deep Space Transponder (DST) where it modulates the downlink carrier. That signal then modulates a low level X-band carrier which is subsequently amplified by the Traveling Wave Tube Amplifier (TWTA) and muted to onc of the three antennas.

Transmission received from the ground operations team is routed to the DST w'here the carrier is tracked. The demodulated subcarrier is detected by the Command Detector Unit (CDU). The CDU searches and finds the acquisition sequence code in the command data, then sends an 'in-lock" inclination and the command data to the Information System.

An Ultra-Stable oscillator (USO) provides an alternative frequency reference to the DST which either relics on the uplink can icr frequency during "two-way" communication or it's own internal oscillator when Operating "one-way".

The HGA is composed of a primary reflector, a subreflector, and individual feeds for each of the frequencies to be used, in addition to the X-band communication frequency, the HGA also supports the following science bands: S-band radio science, S-band Huygens *Probe* science, Ku-band RADAR, and Ka-band radio science. The HGA has a 4-meter symmetric primary reflector with X, Ka, and Ku-band feeds positioned at its apex, A subreflector that is reflective at these three bands is mounted above the primary reflector. An S-band feed horn is positioned behind the subreflector, near what would be the focal point of the primary reflector, and low-gain antenna number 1 (LGA-1) is mounted directly atop the S-band feed The HGA and its associated equipment are boresighted along the spacecraft -Z axis

3.1.6 Payload

The Orbiter design was significantly driven by the requirements imposed by accommodation of the science payload, The description of these instruments and their scientific investigation is beyond the scope of this paper, Table 6 summarizes key requirements.

Instrument	Mass (kg)	Peak" Power (W)	Max. Data Rate (k b/s)	Electromagnetic or Equipotential Requirements	Pointing Requirements	Field of View
Optical Remote Sensing						
1ss	61:1	70.8	266,0	n/a	2 mRad	0.4° 3.5°
VIMS	40.1	29.2	183.0	n/a	2 mRad	1.9° 2.4°
UVIS	16.0	14.0	31.0	n/a	2 mRad	3.7°
CIRS	44,3	32.5	6,0	n/a	2 mRad	0.4° x 0.25°
Microwave Remote Sensing						
RADAR	56.7	120.0	365.0	n/a	2 mRad	n/a
RFIS	15.5	89.0	0.0	n/a	2 mRad	n/a
Field, Particles, Waves						
CAPS	2.14	21.0	16.0	< 5 nT, 10 volt	20 mRad	20" X 160° 10° X 160°
CDA	16.7	19.5	0.5	n/a	175 nuad	100° 180°
INMS	13.0	31.0	1.5	n/a	n/a	16°
MAG	9.8	13.0	2.0	<0.2 nT	n/a	n/a
MIMI	28.5	26.1	8.0	n/a	n/a	90° x 120" 5.2° x 156°
RPWS	29.0	18.3	370.0	<1 μV/m @ 100 Hz <5 nV/m @ 20 kHz	n/a	n/a

'1'able 6 Key Payload Requirements

The science payload is allocated 351 kg and 228-314 W of power ([he power allocation is mode dependent). The Orbiter supplies a gas purge system, providing protection to the contamination sensitive instruments during pre-launch test and integration activities. All exterior Orbiter surfaces are conductive and electrically connected 10 chassis to provide an equipotential surface. The sensitivity of the RPWS to electromagnetic interference led to a system wide synchronization of the power converters to a single, stable 50 kHz source, thereby segregating the spacecraft noise sources to harmonics of 50 kHz. The sensitivity of the MAG to magnetic interference led to the system implementat ion of routing all electrical power and return lines in a twisted wire configuration and to the use of magnetic shielding and compensation magnets.

3.1.7 Thermal Control

Thermal control is required to maintain the spacecraft assemblies and subsystems within acceptable temperature ranges. This is achieved by a combination of spacecraft design features and spacecraft operations.

The design achieves thermal control of spacecraft equipment by the use of insulation blankets (To minimize heat gradients and losses, as well as to protect the equipment during periods when the HGA is pointed off-Sun), louvers (to provide thermostatic control of heat loss from the electronics bays), configuration, and heaters (both electrical heaters and Radioisotope Heater Units). The design relies on close coupling of the thermal energy within the spacecraft bus and primarily on passive thermal control measures, Active control is available in which temperature sensors are used by the information System to cycle the power applied to electrical heaters. This closed loop system allows flight programmable changes to the temperature control range for selected assemblies.

Spacecraft operational constraints provide key aspects of temperature control during the cruise phase. For example, by maintaining the spacecraft with the HGA oriented toward the Sun throughout most of the flight, maximum isolation from solar heating is assured (using the high-gain antenna as a shade.) The HGA will shade all spacecraft elements except for the RTGs, the thruster clusters, the deployed Langmuir probe, the magnetometer boom, and the RPWS antennas.

3.1.8 Fault Protection System

The Fault Protection System (FPS) design is in response to the Project requirement that "No credible single point failure shall prevent attainment of the objectives . ..." and the fact that this requirement leads to a proliferation of redundant hardware elements on the subsystems. The FPS provides sustained spacecraft functions in the presence of faults through the detection, identification, and isolation of a failed element and a switch-over from the failed element to it's standby counterpart.

For most of the mission the FPS will assure only basic spacecraft functions (e.g., commanding, telemetry, thermal control, power management, basebody pointing). For critical activities such as launch and orbit insertion at Saturn, the FPS must also maintain the integrity and correct operation of the on-board sequence in the presence of a fault

Some FPS actions are hardware based, One example is the detection of a low power bus voltage and the response of shedding non-critical (to spacecraft basic functionality) power loads.

Most FPS detection and responses are based in software. In the Guidance and Control System, the flight processor preserves the attitude control function in the presence of sensor, thruster, and gryo failures, The initial line of defense occurs as the software based "hardware managers" test the reasonableness of the data collected. Failed tests here can lead directly to commanded switch-overs. If the fault defies detection at this level, higher level tests are continually performed to check for out-of-bounds pointing and rate performance. Failures of tests at this level lead to a sequence of redundancy management actions that attempt to find the failed element. Finally, if the fault is in the Guidance and Control processor or it's software, the "handshake" with the Information System will cease and the latter will initiate a switch to the redundant processor and/or bring the backup software load on-line.

For FPS functions that are more system wide (e.g., power resource management, thermal balance, ground commandability) the Information System hosts the fault detection, analysis, and corrective response algorithms. The detection sources are of various types (e.g., a signal from the Power System indicating it has just performed load shedding, a temperature sensor measurement has exceeded a high or low bound, or the fact that ground commands have not been processed for an abnormally long time). The resulting recover actions often affect more than one subsystem. The response must be robust enough to be compatible with any initial spacecraft state, and must be tailored to observe power margin, thermal balance, and other spacecraft related constraints,

5.0 Summary

The Cassini Orbiter design results in a very capable and complex system. These characteristics arise from:

1. Supporting the requirements of a large, diverse, rind performance demanding payload.

2. Accommodation of the environmental dynamic range resulting from the planetary swing-bys of Venus to the orbital tour at Saturn

3, Meeting a strict single fault tolerant design requirement.

4. Being sufficient autonomous to require only minimum in ground operations support

6.0 Acknowledgments

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